NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

TECHNICAL NOTE

No. 1087

LANGLEY FULL-SCALE-TUNNEL INVESTIGATION OF THE FUSELAGE BOUNDARY LAYER ON A TYPICAL FIGHTER AIRPLANE WITH A SINGLE LIQUID-COOLED ENGINE

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SUMMARY

An investigation has been made in the Langley full-scale tunnel to determine the thickness and shape of profile of the boundary layer on the fuselage of a typical monoplane fighter airplane with a single liquid-cooled engine. The results showed that, for the range of angles of attack and fuselage stations investigated, the maximum displacement thickness was nearly 1.2 inches and was at the most rearward station (81.6 percent of the fuselage length). The displacement thickness was found to be greatly affected by the pressure gradients over the windshield-canopy combination and in the wing-fuselage juncture. An average value for the shape parameter (ratio of displacement thickness to momentum thickness) between 1.3 and 1.4 was obtained for the turbulent boundary layer.

INTRODUCTION

The design of efficient charge-air and cooling-air inlets for locations where the boundary layer is of appreciable thickness is generally complicated by the tendency of the boundary layer toward separation in the range of inlet-velocity ratios normally encountered in high-speed or cruising flight. In some designs, particularly those in which the inlet is located in a region of adverse pressure gradient or in which the inlet is flush with the fuselage surface, the pressure losses resulting from flow separation are so large that it is usually necessary to dispose of the boundary layer by means of an external gutter or an internal bypass duct. Some idea of the quantities of air that must be removed in order to obtain smooth flow with good pressure recoveries can be obtained from reference 1. As an additional aid to the design of slots, gutters, and other boundary-layer-removal devices, an investigation has been made at the Langley full-scale tunnel to determine the thickness and shape of profile of the boundary layer at various locations on the fuselage of a typical monoplane fighter airplane with a single liquid-cooled engine. The investigation was made on the model without a propeller and over an angle-of-attack range from -1.7° to 4.8°, which corresponds to airplane attitudes ranging from the dive condition to the condition for maximum rate of climb.

SYMBOLS

u	local velocity inside boundary layer
U	local velocity outside boundary layer
U_o	free-stream velocity
P	static-pressure coefficient $\left(1 - (U/U_o)^2\right)$
α	angle of attack of fuselage thrust line with respect to relative free-stream direction, degrees
δ	full thickness of boundary layer
δ^*	displacement thickness of boundary layer $\left(\int_0^\delta \left(1 - \frac{u}{U}\right) dy\right)$
θ	momentum thickness of boundary layer $\left(\int_0^\delta \frac{u}{U} \left(1 - \frac{u}{U}\right) dy\right)$
H	boundary-layer shape parameter $\left(\frac{\delta^*}{\theta}\right)$
y	distance normal to fuselage

MODEL AND TESTS

A full-scale model of a typical liquid-cooled-engine midwing fighter airplane was used for the investigation. The airplane is shown mounted in the Langley full-scale tunnel in figure 1, and the general arrangement of the model is shown in figure 2. As these figures show, the tests were made for the model without a propeller, without ducts, and without tail surfaces. All gaps on the fuselage, such as those left by the removal of the duct and tail assemblies and the gap between the spinner and fuselage, were sealed and faired, and all protuberances, such as radio antennas and gun-blast tubes, were removed. The wing section is a modification of an NACA 230-series airfoil and varies from 15-percent thickness at the root chord to 9-percent thickness at the tip.

The boundary-layer profiles were determined at five fuselage stations ranging from 14.9 to 81.6 percent of the fuselage length (see fig. 3) by means of four rakes mounted normal to the surface at the top, bottom, and two sides of the fuselage, respectively. The rakes (detailed in fig. 4) were $9\frac{1}{2}$ inches in height, consisted of 13 total-pressure and 2 static-pressure tubes each 1/16 inch in outside diameter, and were mounted with the bottom total-pressure tube approximately flush with the fuselage surface. Previous investigations (references 2 and 3) have shown that flow separation in the boundary layer ahead of an air inlet located in the thin boundary layer at the nose of the fuselage occurs at inlet-velocity ratios below 0.3 and that the total-pressure losses are usually small. For this reason, no attempt was made to determine the profiles of very thin boundary layers. In order to prevent any interference effects resulting from rakes installed in tandem, the tests were restricted to the measurement of boundary-layer profiles at a single station at a time.

The investigation was made at angles of attack of -1.7°, 0.2°, and 4.8°, which correspond approximately to the dive, high-speed, and climb attitudes, respectively, for this airplane. All pressure measurements were made at a tunnel airspeed of approximately 63 miles per hour, which corresponds to a Reynolds number, based on a mean geometric chord of 5.47 feet, of 3,200,000.

RESULTS AND DISCUSSION

The results of the fuselage boundary-layer investigation are presented in figure 5 in the form of boundary-layer velocity profiles. An approximate indication of the pressure distribution over the fuselage may be obtained from figure 6. The results indicate that the full thickness of the boundary layer δ at any point on the fuselage ahead of the wing leading edge (ahead of station B) never exceeded 1 inch, but that beyond this point δ began to increase rapidly and was greatly affected by pressure gradients in the wing-fuselage juncture and over the canopy-windshield combination.

For the design of boundary-layer-removal devices, the displacement thickness of the boundary layer δ^* is a more useful criterion than δ because it is more accurately defined by the experimental measurements. The displacement thickness is physically a measure of the displacement of the potential flow resulting from the velocity deficiency within the boundary layer. The exact amount of air that must be removed to ensure efficient inlet performance is not known, but references 4 and 5 indicate that the quantity per unit slot length probably should not exceed $U\delta^*$. Analysis of the results reported in reference 1 indicates that, for correctly designed boundary-layer-removal ducts, good pressure recoveries were obtained in the main duct of a protruding scoop when the quantity of air removed was equal to $0.75U\delta^*$. Curves of the growth of δ^* along the fuselage of the model used in this investigation are given in figure 7.

In general δ^* increased slowly to station B, where it was about 0.1 inch on the top and bottom of the fuselage and 0.2 inch on the two sides. Beyond this station, δ^* was greatly affected by the pressure gradients over the canopy-windshield combination and in the wing-fuselage juncture. Figure 7 indicates that the boundary layer on top of the canopy was very thin, partly because some of it was swept off to the sides of the windshield and partly because the pressure gradient was favorable over the windshield-canopy combination. In the wing-fuselage juncture (fig. 7, right side and left side), δ^* appears to have been considerably increased. This increase is caused by the fact that adjacent boundary layers on the wing and fuselage flow into the high-velocity, low-pressure region in the forward part of the juncture and there is a very steep adverse pressure gradient in the rear part. The smaller values of δ^* obtained on the sides of the fuselage at station D are attributed to the fact that the path of the flow is not directly from the rake at station C to the rake at station D and therefore the values are not for the same streamlines. For the range of angles of attack and stations investigated, a maximum value of δ^* of nearly 1.2 inches was obtained at the most rearward position, station E.

A plot of the shape parameter H, which is an index of the tendency of the turbulent boundary layer toward separation, is given in figure 8. Too much significance should not be attached to the values of H at station A and on top of the fuselage at station C, inasmuch as the boundary layer at these locations was thin and the boundary-layer profiles at these stations were not accurately determined. The average value of H for the turbulent boundary layer was about 1.3 at station B and generally increased slightly toward the rear of the fuselage. The average value of H for all stations was between 1.3 and 1.4. The variation of H with angle of attack was small and inconsistent. Because past tests appear to indicate that the boundary layer will separate when the value of H is between 1.8 and 2.6 (reference 6), separation on the fuselage of this airplane does not appear imminent.

CONCLUSIONS

As a result of the investigation of the boundary layer on the fuselage of a model of a typical monoplane fighter airplane with a single liquid-cooled engine, it was found that:

1. For the range of angles of attack and fuselage stations investigated, the maximum displacement thickness of the boundary layer was almost 1.2 inches at the most rearward station (81.6 percent of the fuselage length).

2. The favorable pressure gradient over the windshield-canopy combination thinned the boundary layer on top of the canopy, and the adverse pressure gradient in the wing-fuselage juncture greatly increased the displacement thickness toward the rear of the juncture.

3. For all stations, the values of the turbulent-boundary-layer shape parameter (ratio of displacement thickness to momentum thickness) were between 1.3 and 1.4 and therefore separation did not appear imminent.

Langley Memorial Aeronautical Laboratory
National Advisory Committee for Aeronautics
Langley Field, Va., March 11, 1946

REFERENCES

1. Czarnecki, K. R., and Nelson, W. J.: Wind-Tunnel Investigation of Rear Underslung Fuselage Ducts. NACA ARR No. 3I21, 1943.

2. Nelson, W. J., and Czarnecki, K. R.: Wind-Tunnel Investigation of Carburetor-Air Inlets. NACA ARR, Feb. 1942.

3. Bell, E. Barton, and DeKoster, Lucas J.: A Preliminary Investigation of the Characteristics of Air Scoops on a Fuselage. NACA ARR, Dec. 1942.

4. Gerber, Alfred: Investigation on Removal of the Boundary Layer by Suction. Translation No. 353, Army Air Corps, Materiel Div., Sept. 26, 1941. (From Mitteilungen aus dem Inst. f. Aerod. No. 6, 1938.)

5. Quinn, John H., Jr.: Tests of the NACA 65_3-018 Airfoil Section with Boundary-Layer Control by Suction. NACA CB No. L4H10, 1944.

6. von Doenhoff, Albert E., and Tetervin, Neal: Determination of General Relations for the Behavior of Turbulent Boundary Layers. NACA ACR No. 3G13, 1943.

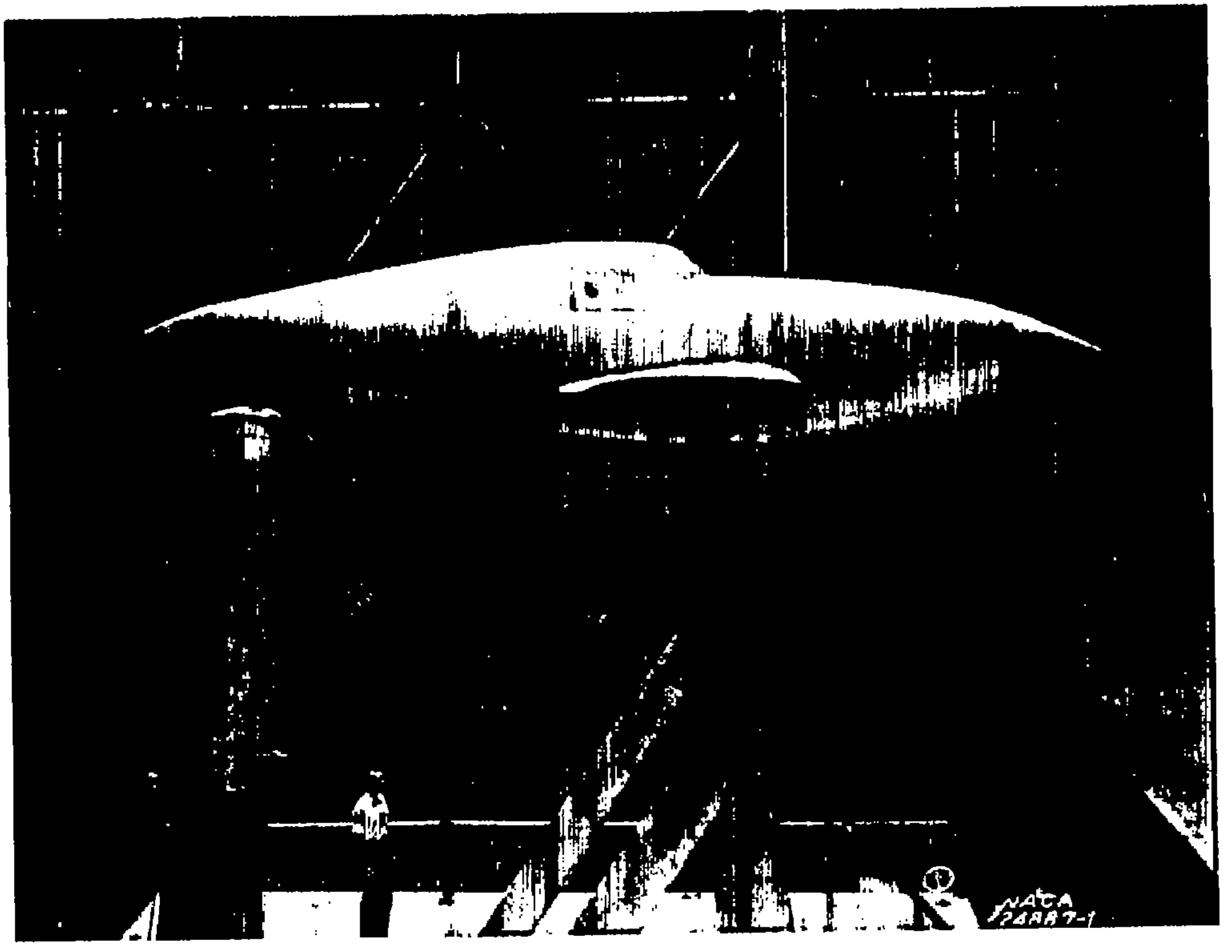


Figure 1.- Side view of model mounted in the Langley full-scale tunnel.

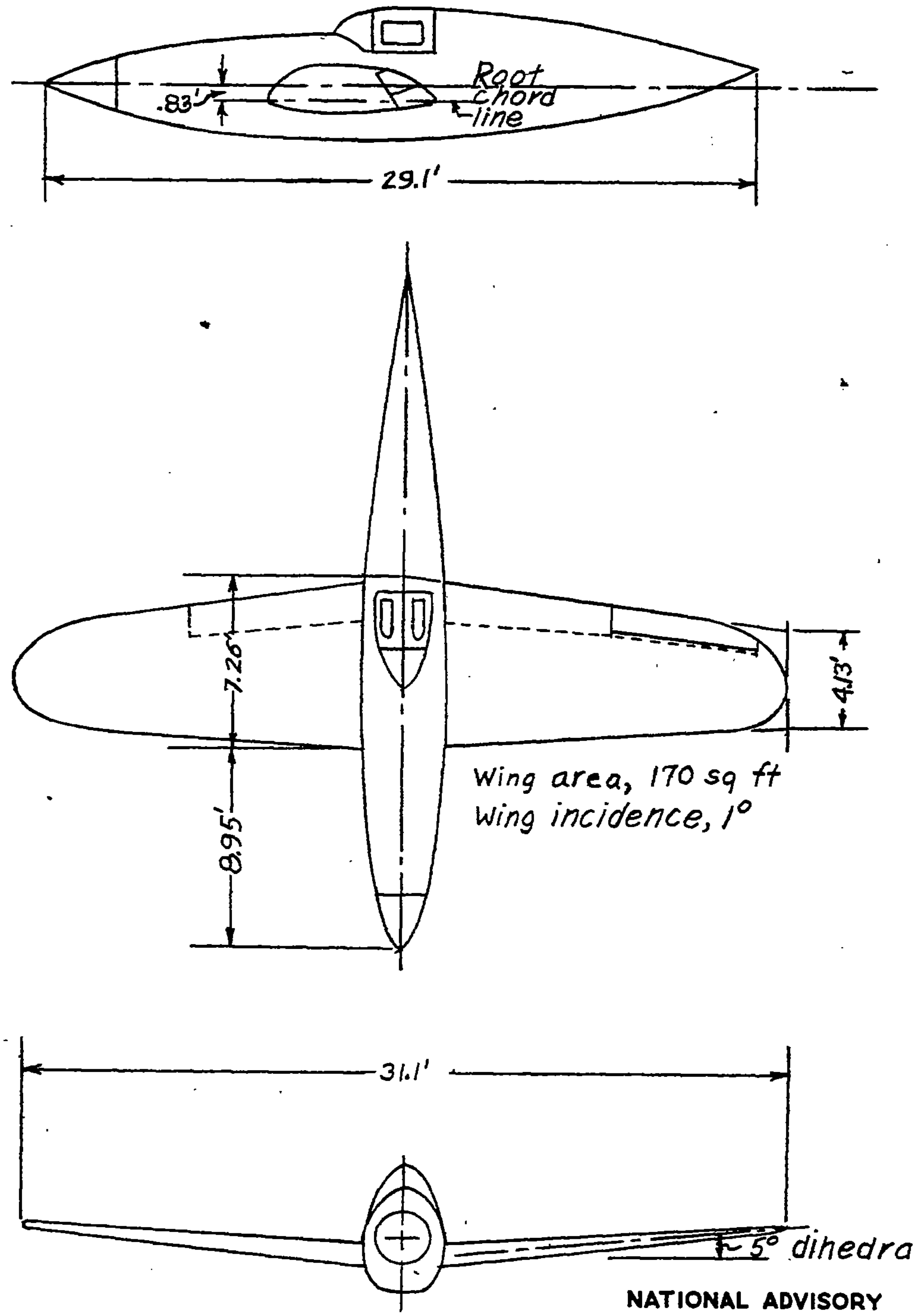
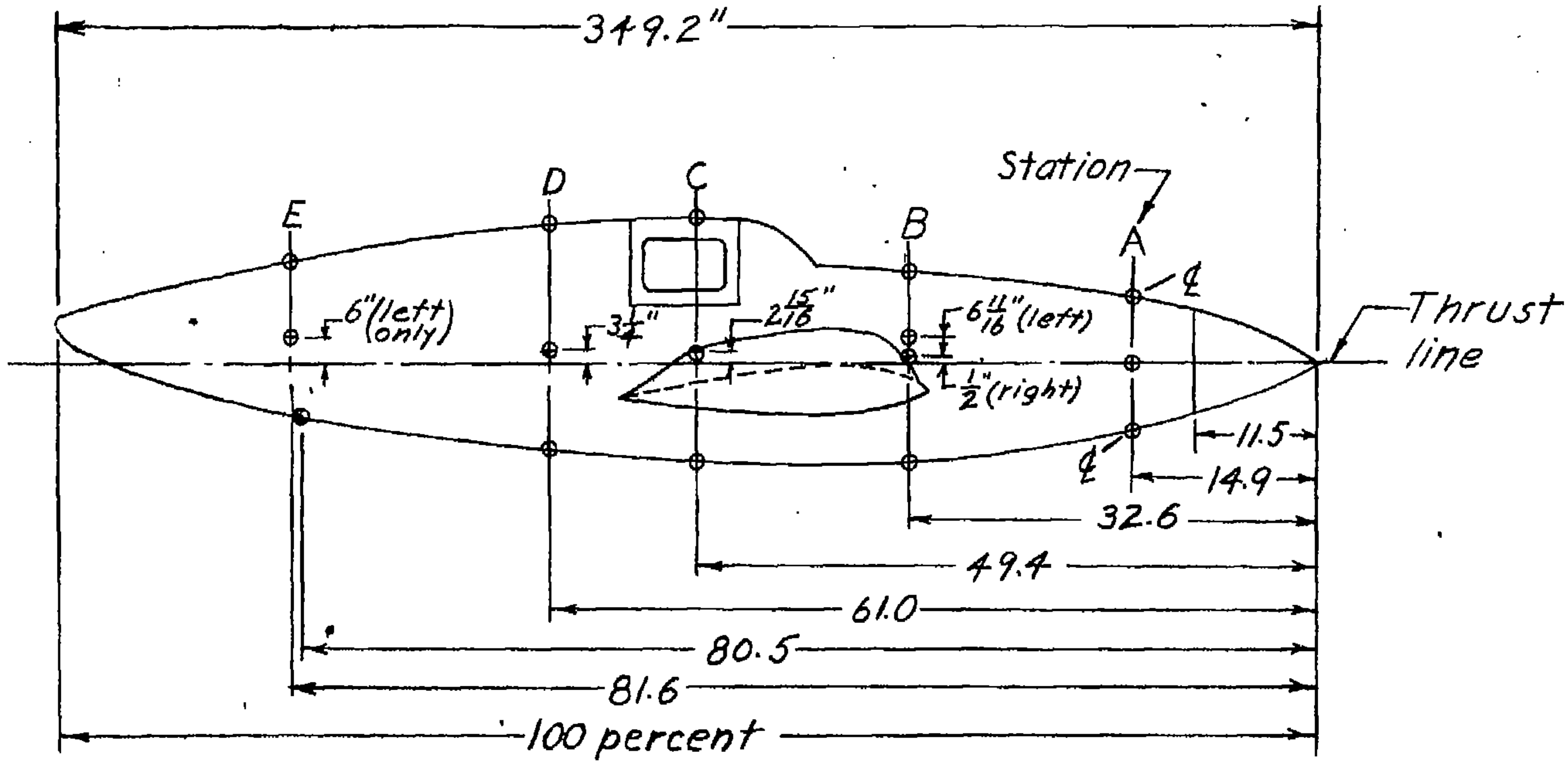


Figure 2.- General arrangement of model.



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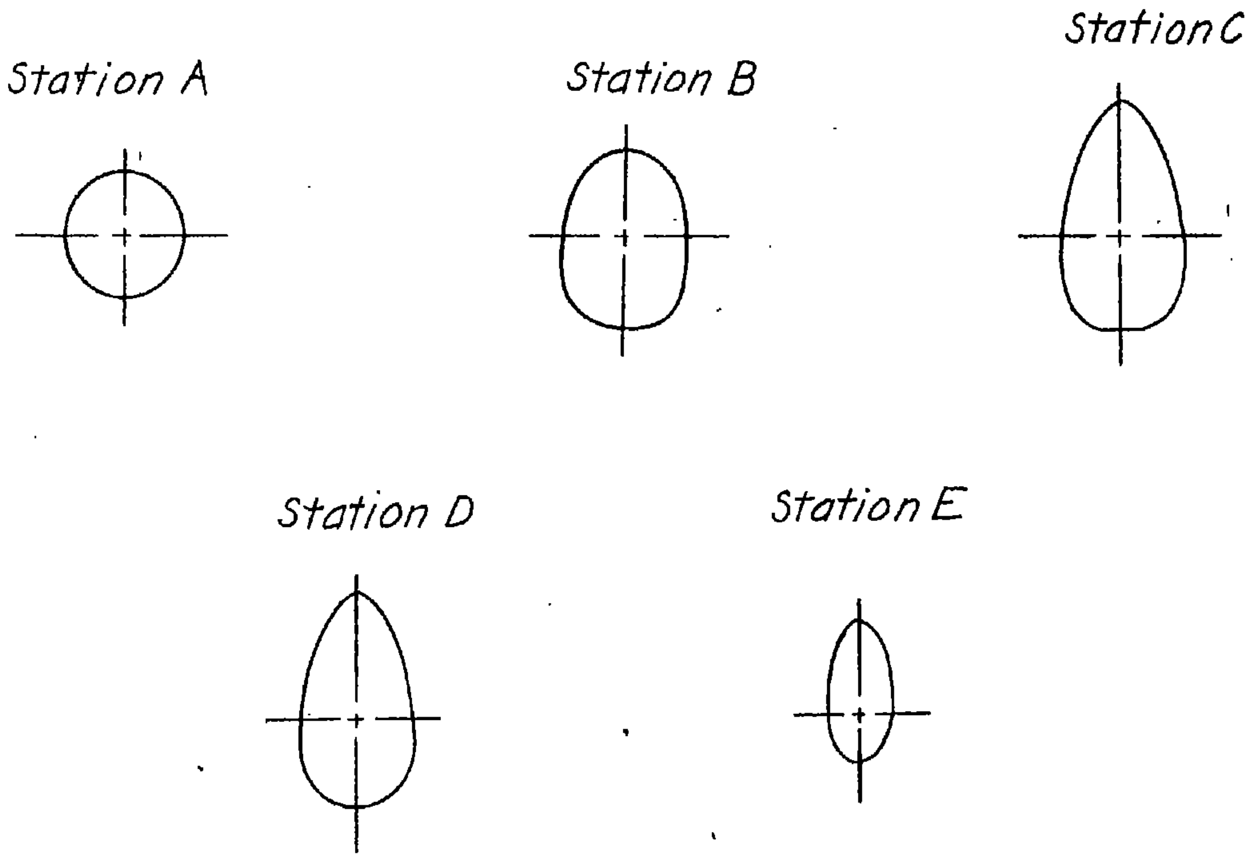
(a) Locations of pressure rakes on the fuselage of the model.

Figure 3.- Details of pressure-rake locations and fuselage contours.



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(b) Contours of fuselage stations.

Figure 3.- Concluded.

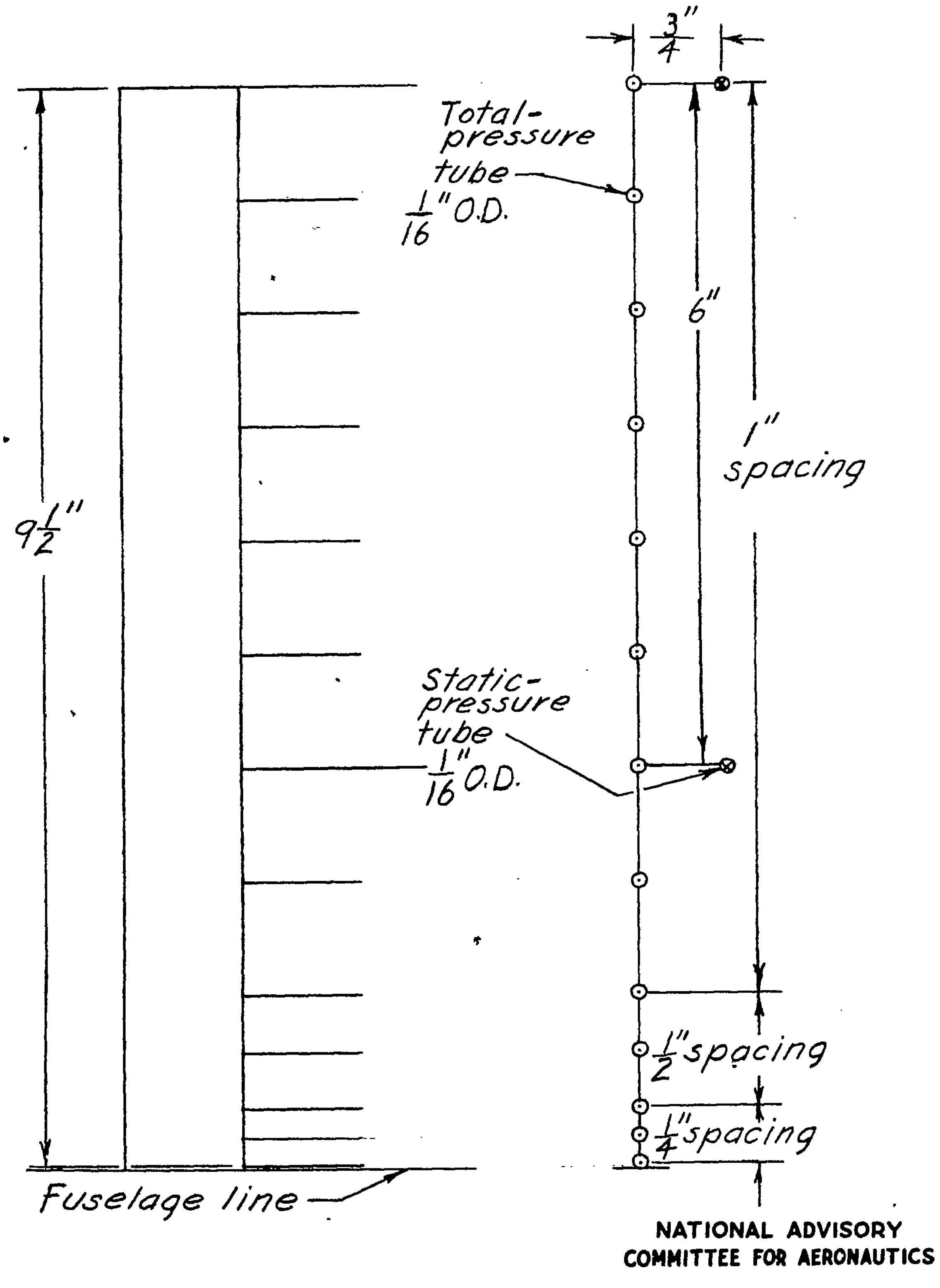
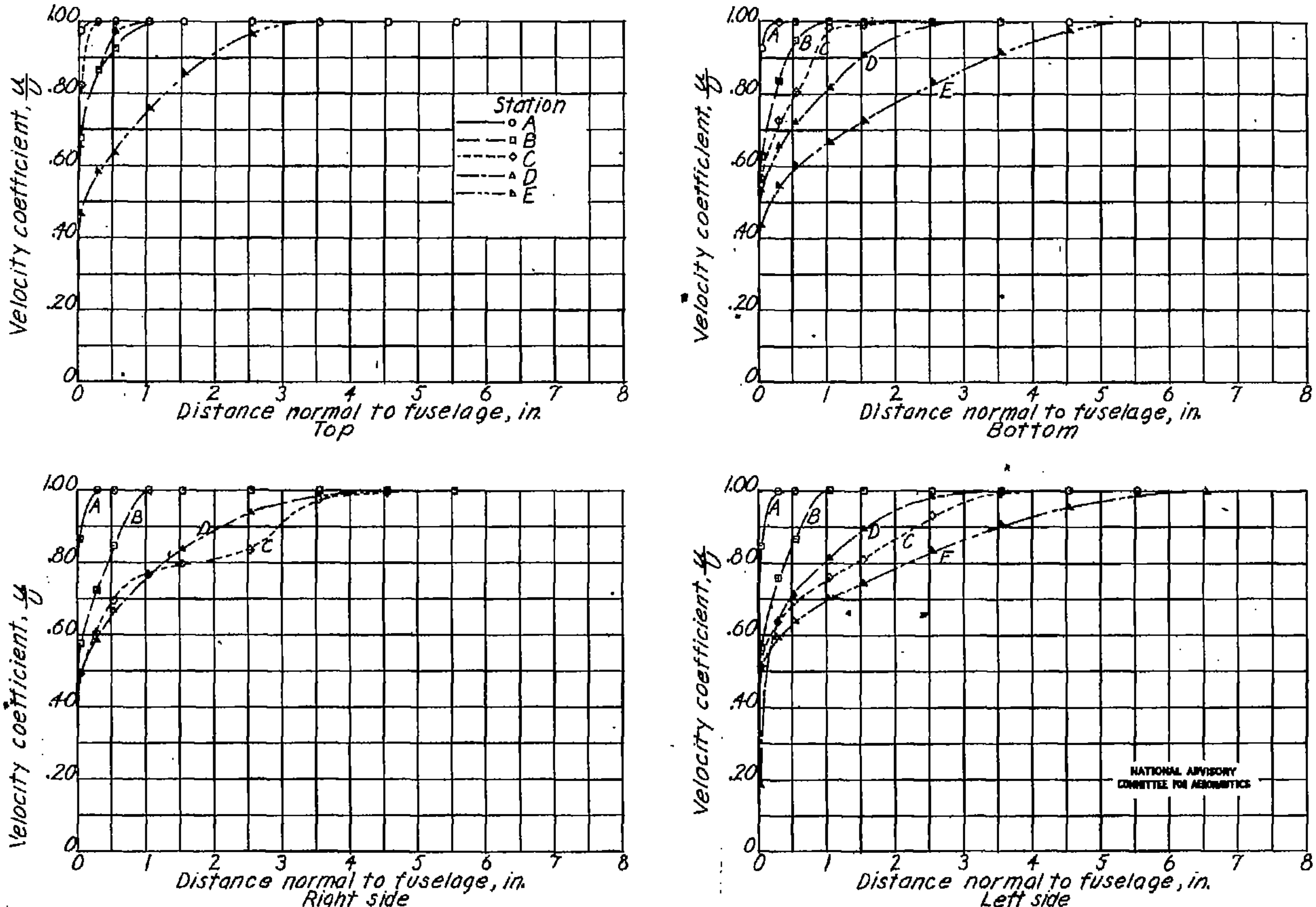
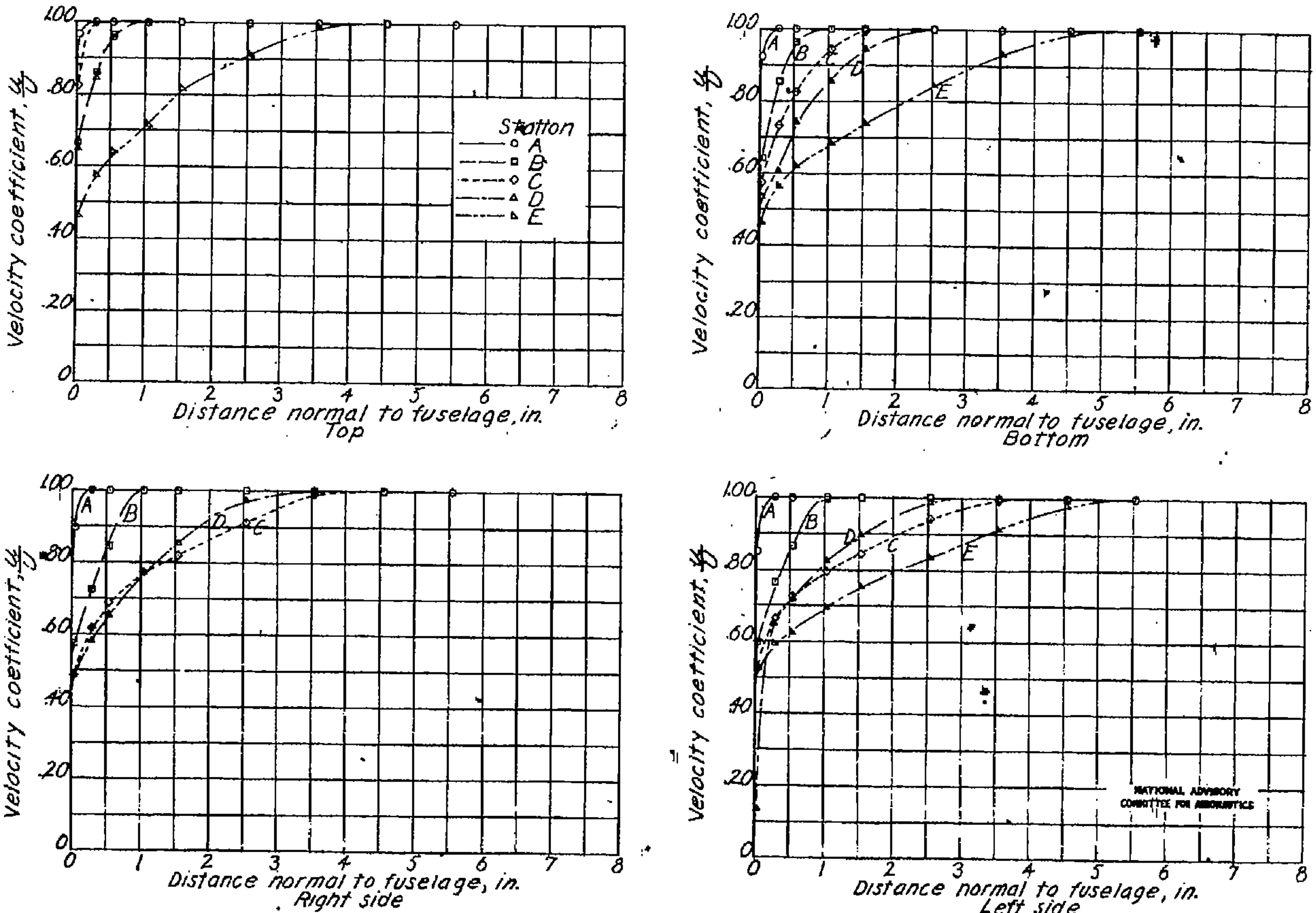


Figure 4.- Rakes used to determine boundary-layer profile.



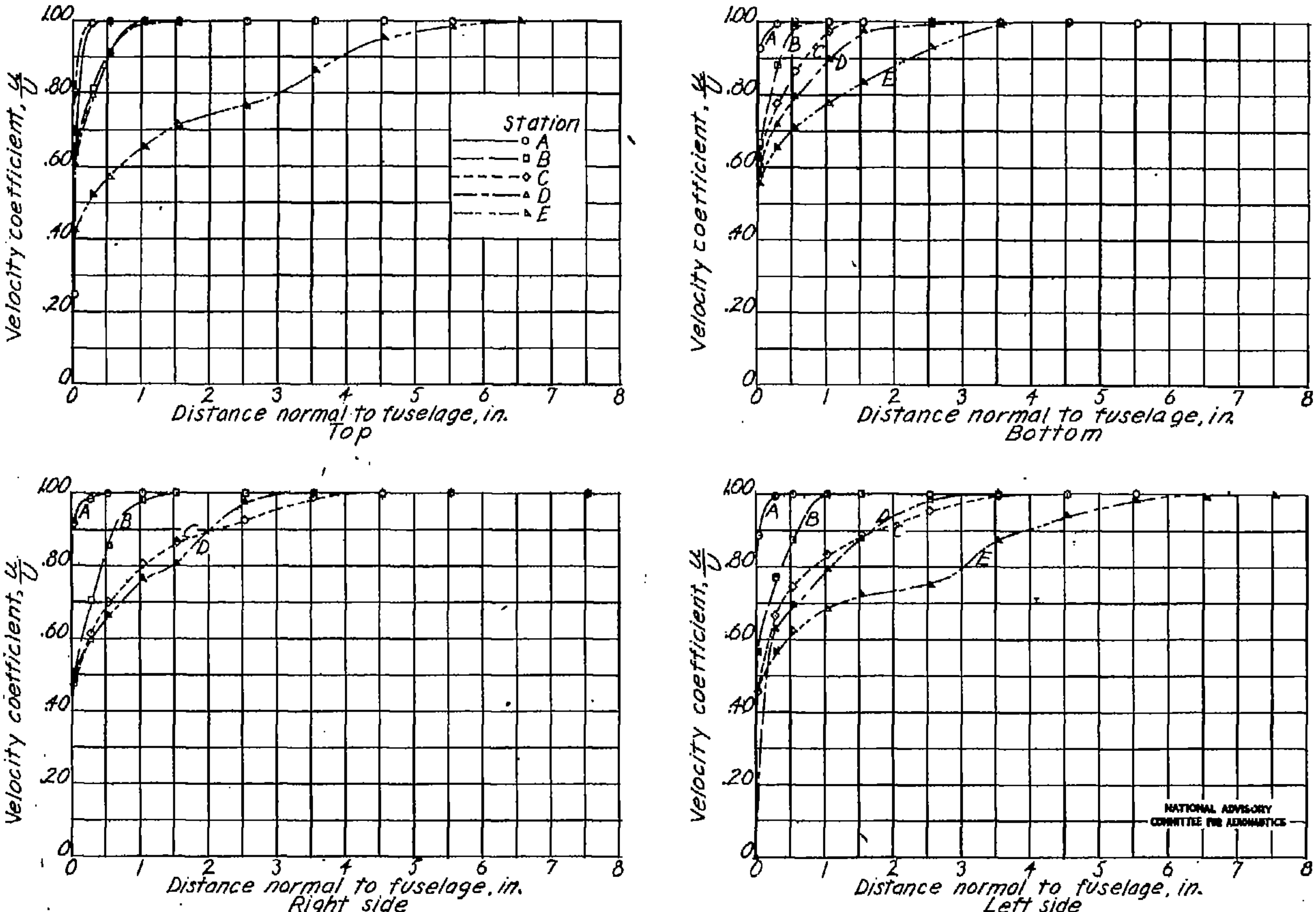
(a) α, -1.7°.

Figure 5.- Boundary-layer velocity profiles over the fuselage.



(b) α, 0.2°.

Figure 5.- Continued.



(c) α, 4.8°.

Figure 5.- Concluded.

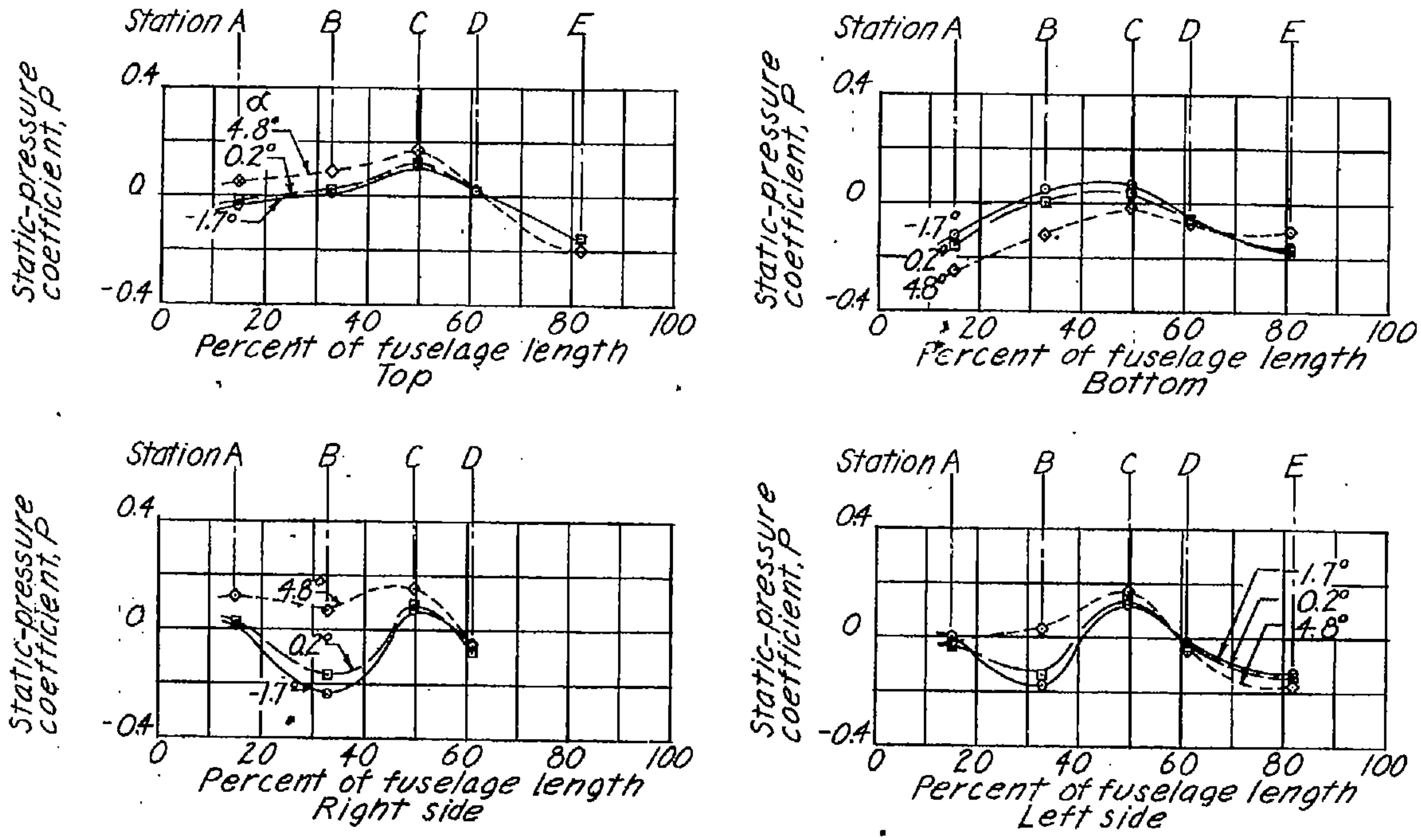


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Figure 6.- Variation of static-pressure coefficient along the fuselage.

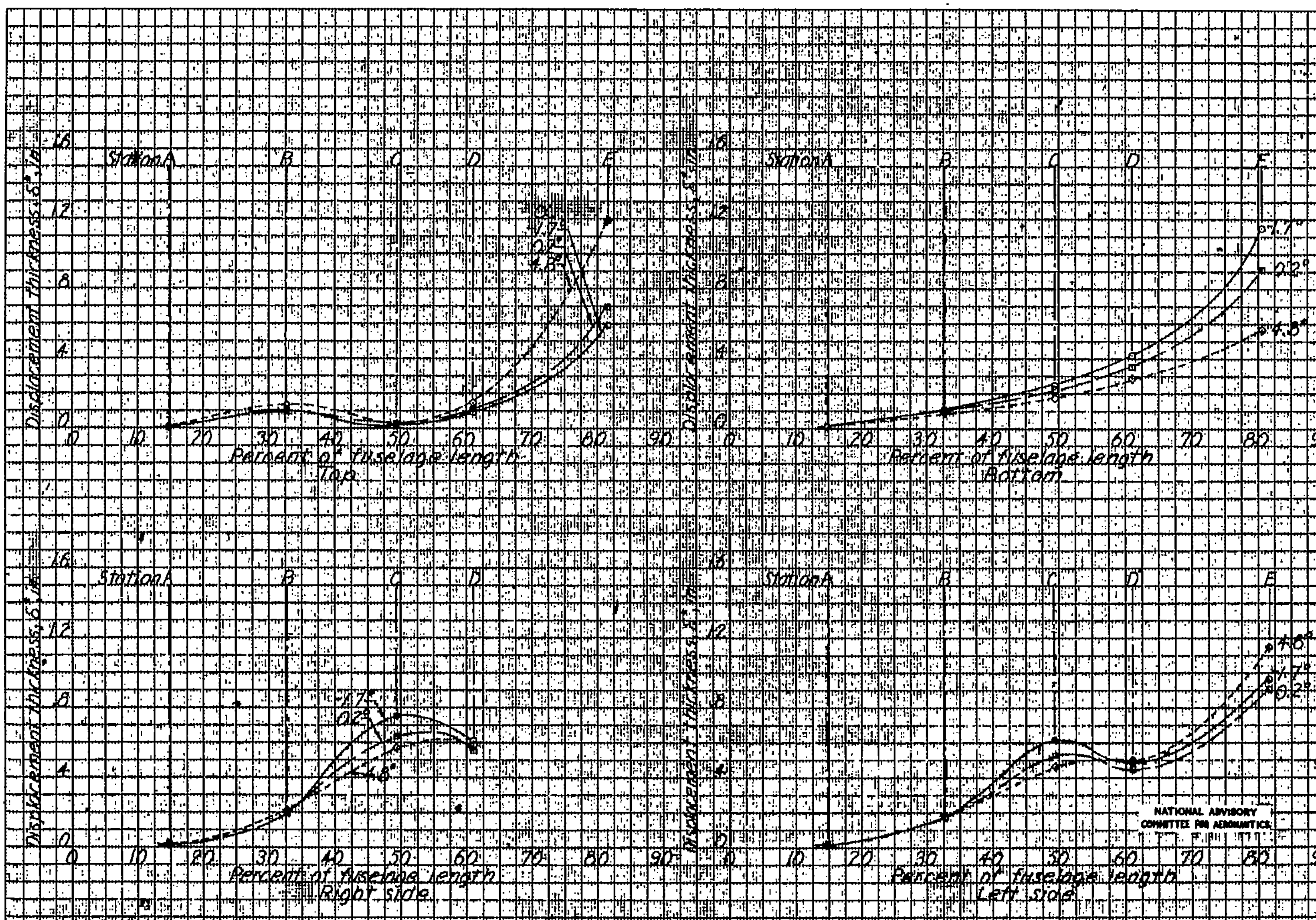

Figure 7.- Variation of displacement thickness δ^*, along the fuselage.

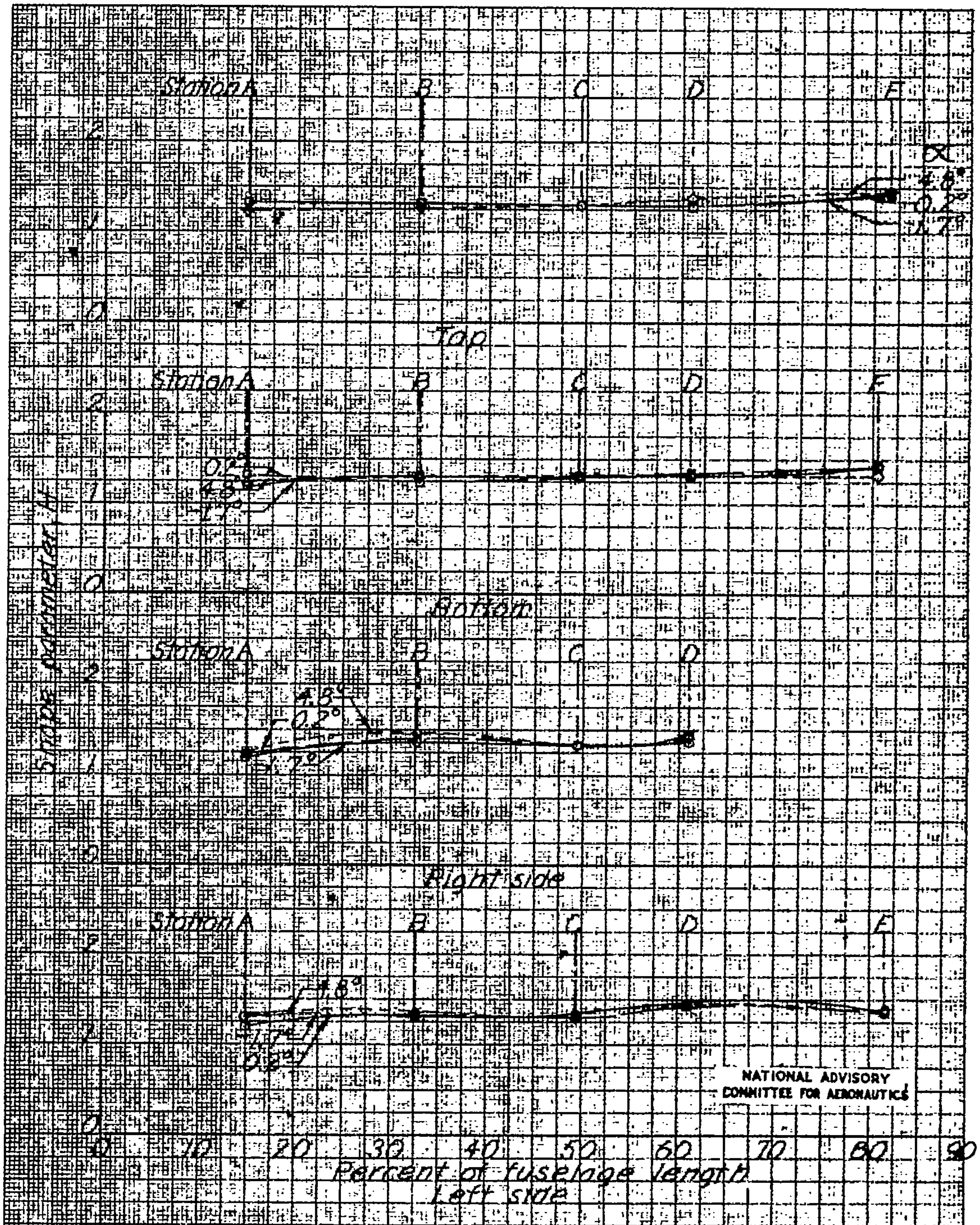

Figure 8.- Variation of shape parameter H along the fuselage.